ACOUSTIC EMISSION DETECTION OF

CRACK PRESENCE AND CRACK ADVANCE DURING FLIGHT

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INTRODUCTION

Since the mid-1970s, several studies have investigated the feasibility of using acoustic emission (AE) to monitor the integrity of aircraft structural components during flight. These studies are completely catalogued in an annotated bibliography of acoustic emission [1,2]. AE has the advantage of being capable of monitoring large components with a single sensor, is truly a passive technique, and can be conveniently used in hard-to-reach locations. Successful development of this technique will offer tremendous savings by reducing the need for major disassembly in order to inspect critical load-bearing components.

The principal problem of acoustic emission monitoring is the unambiguous identification of signal sources (eg, crack growth, crack face rubbing, structural noises). This problem is addressed here, using a multiparameter criterion to identify signals originating at a crack in the presence of airframe noises. To accomplish this, we have developed a data acquisition system specifically for in-flight AE monitoring. This system is stand-alone, is battery-powered, and allows for dual-channel multiparameter processing of the data during flight. This multiple-criterion system greatly enhances the confidence level for the unambiguous separation of crack-related data from airframe noise.

This study also includes the use of an inertially-loaded specimen, containing a well-documented crack, which is attached to the support frame in the instrumentation bay of a Tornado aircraft. Aircraft manoeuvres produce crack advance in the specimen under known g-loading conditions and with superimposed airframe noise. This test apparatus, along with the prototype data acquisition system, has been flight tested in both the Canadian CF-5 and the British Tornado aircraft.

EXPERIMENTAL

The Data Acquisition System

The dual-channel, digital data acquisition system used here was designed and constructed specifically for the recording and interpreting of acoustic emission data during flight. The design is based on criteria derived from the RMC work of almost a decade in the area of acoustic emission monitoring during flight [3,4,5]. These studies established the importance of the difference in arrival time of an event at different locations, signal risetime, and the magnitude and variation of the applied stress at the time of occurrence of the event. All of these parameters are necessary to isolate crack-related events from other noise sources during flight and are recorded by the data acquisition system used here. To provide maximum flexibility, the data acquisition system can be powered either by the aircraft electrical system or by batteries.

The output of each each of the two piezoelectric sensor elements is amplified by an integrated preamplifier (with nominal gain of 40 dB) located inside the sensor casing. The resulting signal is buffered, logarithmically amplified, envelope followed and peak detected. These operations are accomplished using signal conditioning boards custom-made for the purpose (figure 1). The output of each envelope follower is separately fed into the digital data acquisition system where the times of preselected amplitude threshold crossings 6 dB apart are recorded (figure 1). The output of the peak detectors and accelerometer are digitized by an A/D convertor and stored in memory.

All of the above data are compressed into an event record which includes the time of occurrence of the event at each sensor, the difference in arrival times at two sensors (Δt), event risetimes for 6 dB change in amplitude, event durations, event decay times and event peak amplitudes. The resulting data set is then extracted from the data acquisition system via an RS-232 interface and stored on disk on a portable personal computer. Extensive screening of data, field analysis and interpretation can be carried out immediately. Final analysis and interpretation are accomplished using spread-sheet software. Table 1 lists the general specifications of the apparatus.

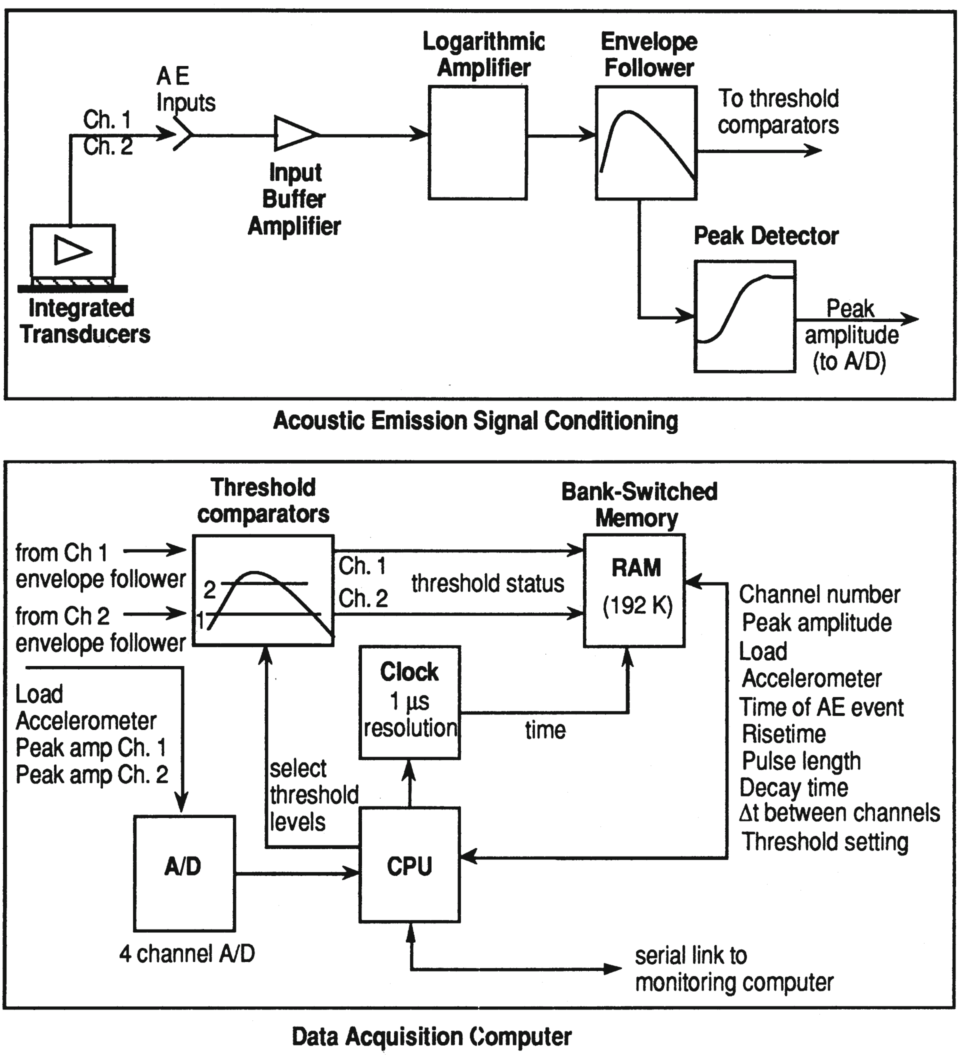


Figure 1 - Schematic diagram of the acoustic emission signal conditioning and data acquisition computer.

Table 1 - General specifications for the RMC digital data acquisition system for in-flight acoustic emission monitoring applications

2 Channels AE	60 dB dynamic range
2 Analog Channels	0-10 V full-scale deflection
Power	10 Watts maximum
Memory	192 Kbyte RAM with battery back-up
Dimensions	23 cm x 13.5 cm x 25 cm
Weight	2 kg
Mass Data Storage	transfer to portable PC via RS232 interface

System Calibration

A detailed calibration of the acoustic emission system (inertial-loading frame, fatigue specimen, sensors and data acquisition apparatus) was carried out for source signals injected at various locations. The source signals used are the helium gas jet, pencil lead fracture, pulsed YAG laser and fracture-related events generated during fatigue crack growth and overload in the laboratory. The measured mean arrival time differences (Δt) and mean risetimes are listed in Table 2 for various locations. These two parameters were extremely efficient at isolating crack-related events from incoming airframe noise, as indicated in table 2 and shown in figure 3. Figure 3 shows the measured results obtained using 0.5 mm diameter pencil fracture as a simulation source.

The Inertially-Loaded Specimen

Figure 2 shows a schematic diagram of the inertially-loaded 7075-T651 aluminum fatigue specimen clamped in the inertial loading frame support. Silicone fluid provides proper acoustic coupling of the fatigue specimen to the loading frame to ensure that airframe noises are transmitted to the fatigue specimen for detection by the acoustic emission sensors. Prior to mounting in the loading frame, the test specimen was precracked to a crack length which would cause crack propagation when the 0.9 kg inertial load was subjected to an acceleration in excess of 3 g. The aircraft acceleration is sensed by an accelerometer (Entran Devices, Inc., model EGD-240) mounted in the loading frame support block. The inertially-loaded specimen was acoustically coupled to the aircraft support frame in the instrumentation bay.

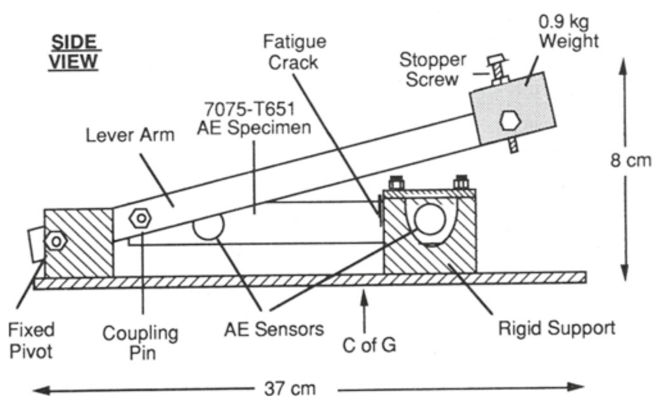


Figure 2 - Schematic diagram of the inertial loading apparatus and precracked 7075-T651 aluminum test specimen.

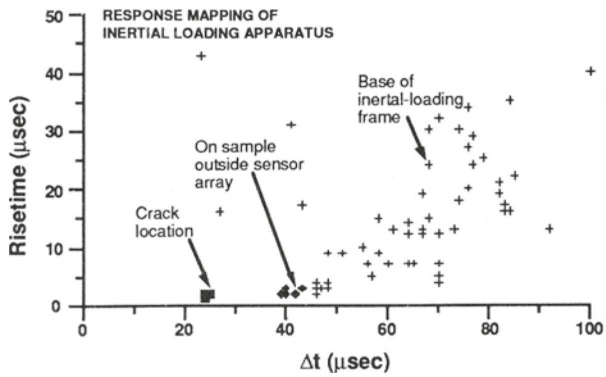


Figure 3 - Measured values of Δt and signal risetime for various locations.

Table 2 - Measured mean values of Δt and signal risetime for various locations.

Position of source	**Δt** (μsec)	**risetime** (μsec)
Crack location	24	2
On specimen, outside sensor array	41	2.5
Base of inertial-loading frame	65	16

The Test Flight Results

Figure 4 shows the test flight profile as measured by the data acquisition system. Each recorded data point corresponds to the detection of an event. These events result from crack advance, crack face rubbing and airframe noises. Included in the flight profile are three aircraft manoeuvres at 15 min 12 sec, 67 min 38 sec and 68 min 30 sec, respectively, relative to take-off. These manoeuvres resulted in successively increasing maximum g values of 2.7 g, 4.3 g and 5.2 g, respectively, and applied a sequence of increasingly large stresses to the fatigue crack through inertial loading. Electron microscopic examination of the fracture surface revealed that an increase in crack face area of 0.63 mm^2 resulted from the 4.3 g_{max} manoeuvre. Fracture of the specimen occurred during the 5.2 g_{max} manoeuvre.

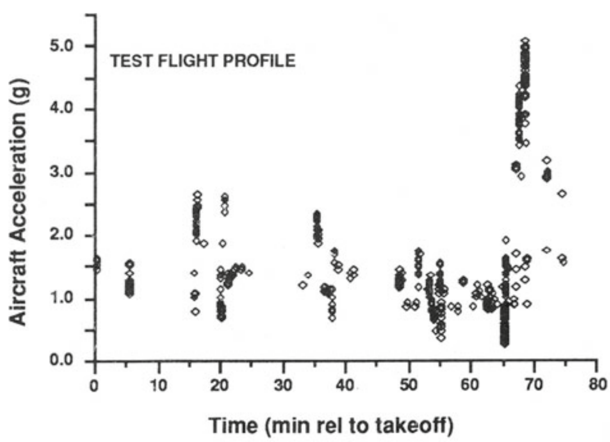


Figure 4 - Aircraft acceleration as a function of time for all of the events detected during flight. These include all sources (crack advance, crack face rubbing and superimposed airframe structural noises).

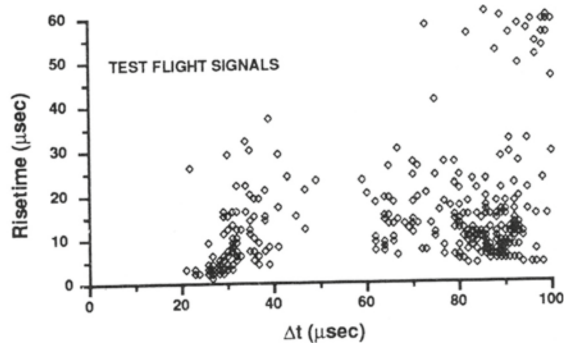
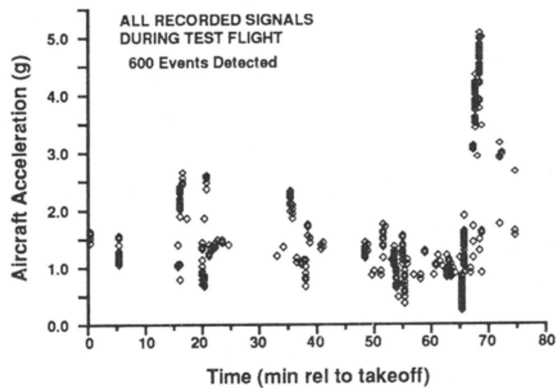


Figure 5 - Signal risetime as a function of difference in arrival time.

Figure 5 shows the scatter plot of signal risetime and difference in arrival time of each acoustic emission event and noise signal detected during the test flight (figure 4) for comparison with the calibration data (figure 3). Comparison of figures 3 and 5 show that the majority of detected signals arrive at the sensors via the base of the inertial-loading frame. Based on this comparison, only those events with $\Delta t = 24 \pm 6$ μsec and risetimes of 3 ± 2 μsec were accepted as crack-related events.

Figure 6 shows the effect of applying these conditions to the data of figure 4. Note that all but the final two high-g manoeuvres are removed from the data set by the dual-parameter filter.



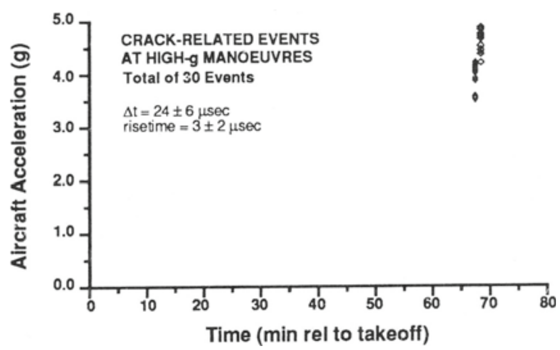


Figure 6 - Comparison of the occurrence of all.events detected during the test flight (upper graph) with the occurrence of events unambiguously identified as crack-related ($\Delta t = 24 \pm 6$ μsec and risetime of 3 ± 2 μsec). Note that these latter events occur only during progressively high-g manoeuvres which provide the stresses required for crack advance.

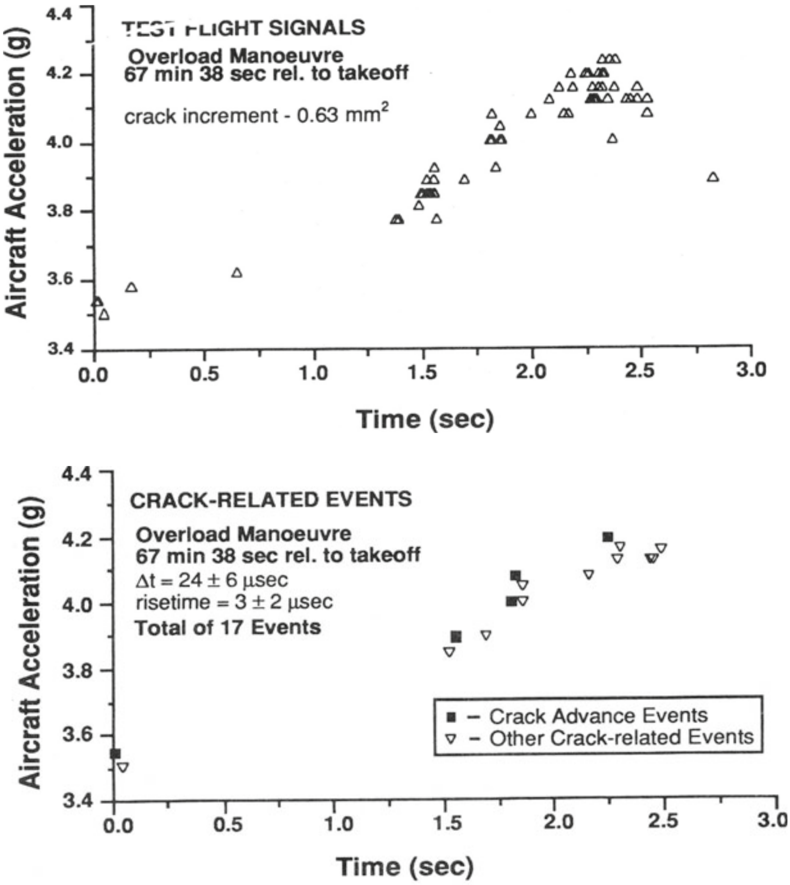


Figure 7 - Crack-related signals detected during the 4.3 g_{max} manoeuvres. The upper graph includes all detected events, while the lower graph includes only those events which are crack-related. The distinction of crack advance events from other crack-related events can be accomplished using the Kaiser Effect.

Figure 7 shows an expanded view of the data recorded during the 4.3 g_{max} manoeuvre which resulted in crack advance. In the lower graph, only the crack-related events are shown. By application of the Kaiser Effect, these events are identified separately as crack advance and other crack-related events.

SUMMARY AND CONCLUSIONS

A precracked 7075-T651 aluminum specimen mounted in an inertial loading apparatus was subjected to stresses large enough to propagate the crack during flight. These crack propagating stresses occurred during specific test flight manoeuvres of a British Tornado aircraft. The specimen and loading apparatus were part of a secondary structure bolted directly to the support frame in the instrumentation bay of the aircraft. Acoustic emission data and aircraft acceleration were recorded using a data acquisition system designed at RMC specifically for in-flight monitoring applications. The crack-related events were isolated immediately following the test flight. More detailed analysis was later carried out to confirm the results and to separate crack advance events from other crack-related events.

The selection of events which are unambiguously caused by crack-related sources was carried out using a very restrictive dual criterion ($\Delta t = 24 \pm 6$ μsec and risetime of 3 ± 2 μsec) derived from specimen calibration and comparison of the test flight data with laboratory crack growth results obtained for 7075-T651 specimens with geometry and sensor configurations similar to that of the specimen used for the test flight. The 30 crack-related events selected in this manner from 600 events detected during the test flight occurred as the result of high-g manoeuvres during which crack advance would be expected. Eleven of the 30 crack-related events are attributed to crack advance by application of a temporary Kaiser Effect criterion. Thus, we confirm the feasibility of the unambiguous detection of crack growth and presence in 7075-T651 during flight, provided that detailed calibration of the structure is carried out.

Future work in this area will include application of the RMC data acquisition and analysis system to the monitoring of airframe fatigue tests, the monitoring of a slowly growing fatigue crack in a secondary structure during flight, and the in-service monitoring of failure-prone airframe structural components during flight.

ACKNOWLEDGEMENTS

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